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DESIGN AND TEST OF A BORON - ALUMINUM HIGH TEMPERATURE WING

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of boron - aluminum advanced composite material in a simple, low-cost spar- rib-skin construction for a thin airfoil structure was investigated for		
high temperature application up to 589 degrees K.		
The design concept developed		on-aluminum skins, to
carry the primary bending and torsion loads, mechanically fastened to a		
light gage steel sub-structure, which resists transverse shear and stabilizes		

UNCLASSIFIED SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered) the skins. The viability of the concept depends on whether this stabilization of the skin material can be accomplished with a practical number and spacing of substructure elements. A weight saving of one third in comparison to the production article is projected in this boron-aluminum version of the BYM-34E wing. A major wing subcomponent was fabricated and static tested to validate the structural adequacy of the overall design.

SUMMARY

The feasibility of utilizing the high buckling stability characteristics of boron-aluminum material in a simple, low-cost spar-rib-skin construction for a thin airfoil structure has been investigated for high temperature application up to 589 degrees K. A weight saving of 30% in comparison to the production article is projected in this boron-aluminum version of the BQM-34E wing, while increasing its temperature capability to 589 degrees K.

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INTRODUCTION

The emphasis in current Naval aircraft structural development is on reduction of weight and cost and improvement of performance. In addition, as flight speeds increase and lift augmentation and thrust vectoring are utilized in Vertical-Short Takeoff/Landing (V/STOL) aircraft, high-temperature structures may be required to withstand the effects of aerodynamic heating and hot exhaust gases. Significant achievements have been made in reducing structural weight by utilizing composite materials, i.e., boron or graphite-epoxy, for moderate-temperature applications, up to 450 degrees K. Similar improvements for higher service temperatures, up to 589 degrees K, require the use of graphite/polymide or boron-aluminum materials.

Despite its current high cost, which is expected to be significantly reduced as usage increases, boron-aluminum has many advantages. It has higher longitudinal stiffness and strength than steel and greater room-temperature transverse and shear stiffness than titanium, while its density is less than that of aluminum. In addition, it has high bearing strength and retains the high thermal and electrical conductivity and weldability of its aluminum matrix.

The objectives of this program were to develop a high-temperature (589 degrees K) composite structural design applicable to thin lifting surfaces, and to demonstrate the concept in a primary aircraft structural component.

Normal design practice for a thin aerodynamic surface, which is being considered here, would be to use full depth honeycomb sandwich construction. However, for high temperature applications, bonding of the skins to the honeycomb core becomes a problem. It was the intent of this program, therefore, to investigate the feasibility of stabilizing the skins with discrete stiffeners at a reasonable cost and weight.

The design which was developed in this program consists of variable thickness boron-aluminum skins, to carry the primary bending and torsion loads, mechanically fastened to a light stainless steel substructure, which resists transverse shear and stabilizes the skins. The viability of the concept depends on whether this stabilization of the skin can be accomplished with a practical number and spacing of substructure elements. Fabrication cost and complexity were minimized by using simple shapes and conventional metal forming and fastening methods. The demonstration article chosen is the wing of the BQM-34E remote-piloted vehicle whose maximum thickness is only three percent of its chord, Figure 1.

Information from material and structural tests has been utilized in the evolution of the wing design. Experimentally verified material stiffness and strength properties have been incorporated into the analysis, together with buckling criteria which have been modified as a result of subcomponent development tests.

DESIGN REQUIREMENTS AND CONSIDERATIONS

The design of the B/Al version of the BQM-34E wing is based on production wing static strength, stability and flutter requirements. The critical flight load condition dictating the design, results from a 5g symmetric pull-up at R.T. An additional design requirement, a 4g symmetric pull up at 589°K, was specified for the B/Al prototype wing.

The high temperature requirement necessitated the selection of thermally compatable materials to be used in the wing design. Specifically, the coefficient of thermal expansion for the light gage metal supporting substructure had to closely match that of the B/Al skins to minimize thermal stresses at elevated temperatures. Stainless steel (TH1050) which is structurally adequate at 589° K and thermally compatible with the B/Al laminate skins was selected as a satisfactory material for the substructure. Both materials have a thermal expansion coefficient of approximately $11.0\,\mu$ m/m $^{\circ}$ C.

Stiffness requirements dictate that the wing exhibit flutter free behavior in the flight regime ranging from Mach 1.1 at sea level to Mach 3.0 at 23600 m (60000 ft.).

B/A1 WING - FINAL DESIGN OVERVIEW

WING CONFIGURATION

The profile of the B/Al version of the BQM-34E wing duplicates that of the production metal wing. A low cost design approach was followed by approximating the actual wing aerodynamic contour with a simplified wedge shape. Referring to Figure 2, all chordwise wing sections are constant depth closed out with simple wedge leading and trailing edge pieces. Spanwise, the wing tapers linearly from root to tip. Across the center wing box the skins are allowed to assume their natural pure bending curvatures.

SKINS

The basic skin configuration for the B/Al wing design, shown in Figure 3, consists of B/Al tension and compression skin pieces with tailored $(0^{\circ}, \frac{1}{2}45, 90^{\circ})$ ply construction. Because of the B/Al laminate fabrication diffusion bonding process which involves a multi-step pressing operation, the B/Al main wing skins were kept to a manageable size by incorporating a wing center line skin splice. The joining is accomplished with a single stainless steel splice plate (2.54 mm, (.1 in.)) and a double row of mechanical blind fasteners (4.76 mm (3/16 in.)). Also, separate B/Al trailing edge pieces and stainless steel sheet leading edge pieces are spliced to the main skins along substructure spars.

Both main skins are step tapered, with gradual ply build up toward the wing centerline, optimized to satisfy critical flight load requirements. The final laminate design for the skins was arrived at through iterative stress analysis and experimental specimen and subcomponent testing. The final ply scheme for the tension and compression B/Al skins is schematically

shown in Figures 4 and 5. Skin laminate design drawings are attached at the end of the report. Both tension and compression skins are four plies (.108 cm) at the wing tip, with ply build up to 13 plies (.352 cm) and 16 plies (.168 cm) respectively, across the overall wing box. The extra plies are added to the compression skin to satisfy buckling requirements. Also, both skins are locally built up to 24 plies (.640 cm), in the area of high stress adjacent to the aft attachment of the wing.

SUBSTRUCTURE

Considering only half the wing, referring to Figure 6, the main elements of the light gage stainless steel substructure include seven spars, a tip and root rib and five wing/fuselage bolt attachment fittings. The spar and rib elements are mainly channels, with gages varying from .052 cm (.020 in.) to .127 cm (.050 in.) depending on design requirements. The spar elements run along constant percent of chord lines and are tapered linearly from wing root to tip. The five wing/fuselage attachment fittings tie the substructure elements together along the wing/fuselage bolt attachment lines. Forward spars extend from the fittings across the wing box. The flanges of the wing box spars are separate angle pieces rolled to match the curvature of the skins. The angles are internally spot welded to web sheets to form channel elements.

FASTENING

Fastening of all the structural elements is accomplished with rivets. Standard stainless steel .476 cm (3/16 in.) dia. solid rivets in conjunction with shear clips are used to fasten the substructure elements together. Fastening of the B/Al skins to the substructure is accomplished with .476 cm (3/16 in.) dia. stainless steel blind fasteners. Double rows of blind fasteners in conjunction with .476 cm (3/16 in.) stainless steel plates, as shown in Figure 7, are used to splice the upper and lower half skins together at the wing center line. Similar splice designs are used to connect leading and trailing edge pieces to the main wing skins.

ANALYSIS

NASTRAN

Stress analysis of the B/Al wing design was accomplished by constructing a finite element model, and running a series of NASTRAN static analyses, for the critical 5g maneuver load condition, optimizing the design. The tension and compression wing skins were modeled with quadrilateral and triangular plate elements which have both inplane and bending stiffness. B/Al laminate constitutive relationships used in the NASTRAN analysis were determined from basic laminate theory using the material property constants of unidirectional B/Al. The substructure spars and ribs were modeled with bar elements with shear properties built in. Because of wing symmetry only half of the wing needed to be modeled. The model configuration including grid point and element identification is shown in Figures 8 through 10.

Bulk data for the NASTRAN model is included in Appendix A. Maximum tension and compression skin limit load stresses obtained from NASTRAN for the final laminate design are shown in Figures 11 and 12 respectively.

BUCKLING ANALYSIS

The boron aluminum wing compression skin was sized to satisfy buckling requirements by using NASTRAN stresses in conjunction with standard orthotropic simply supported plate theory. Since the skins are mechanically fastened to the substructure the simply supported boundary condition is a conservative assumption. Buckling loads were calculated for the most highly stressed compression skin NASTRAN elements in each discrete skin gage region. Several iterative cycles were needed to size the skin for buckling stability. Table 1 lists the final results for the compression skin buckling analysis. The critical buckling load due to compression, Nxcr, and the critical buckling load due to shear loading, Nxycr, are compared with the loading the laminate must withstand at design ultimate, Nxult and Nxyult. Margins of safety in buckling due to combined compression and shear loading were calculated using the relation

M.S. =
$$\frac{2}{R_L + \sqrt{R_L^2 + 4R_S^2}} - 1$$

where:

$$R_{L} = \frac{N_{xult}}{N_{xcr}}$$

$$R_S = \frac{N_{xyult}}{N_{xyer}}$$

Although the margins of safety for ultimate load were slightly negative for several of the compression skin elements, they were considered acceptable at this point since the analysis was conservative and testing was planned to assess the accuracy of the analysis method. Also, when considering design limit loading, all margins of safety would be positive.

DYNAMIC ANALYSIS

A NASTRAN real eigenvalue run was made to obtain normal mode data for the B/Al wing design. Based on the results of this run and the fact that the B/Al wing design is both stiffer and has less mass than the production wing, the wing was assumed to be flutter free and a rigorous flutter analysis of the B/Al wing was not included in the design cycle.

EXPERIMENTAL TESTING

INTRODUCTION

In order to experimentally validate design procedures and establish a design criteria on which to base the final B/Al full scale wing design, a

series of coupon specimens and two major subcomponents were fabricated and tested. The testing phase of the program included only room temperature testing. This was justified because the critical flight load condition is the R.T. 5g maneuver. To save on fabrication cost 4130 steel was substituted for the stainless, in all subcomponent substructural members.

COUPON SPECIMENS

A number of B/Al coupon specimens including tension and rail shear were tested to validate the material properties used in the design of the full-scale wing. The specimen configurations are shown in Figure 13. A summary of the coupon test results run at NADC are shown in tables 2 through 4. Results of tensile specimen tests run by Americom, Inc. on the basic B/Al laminates used in the tension and compression wing skin design are shown in Table 5. Results of these tests were satisfactory, ultimate loads and material properties in some cases were slightly lower than available standard B/Al properties.

BOX BEAM SUBCOMPONENT

Design

In order to evaluate the manufacturing processes intended for construction of the full-scale wing and to verify the buckling capability of the B/Al compression skin, a box beam specimen representative of the aft wing box region as shown in Figure 14 was designed, fabricated and tested. The aft wing box region was selected for experimental investigation because the compression skin is buckling critical in this area and a box beam type specimen presents minimum fabrication complications and can be symmetrically loaded to facilitate testing.

The box beam specimen, shown in Figure 15, which has a span of 107 cm (42 in.) and a width of 18 cm (7 in.) incorporates the same basic design features as found in the actual aft wing box. The detailed engineering drawing of the box beam is included in the foldouts. The box beam center span between the attachment bolt hole center lines, like the actual wing, is 45.7 cm. (18 in.). The center span substructure channels are constructed of 7.62 mm (.030 in.) rolled 4130 steel angles, to form constant radius flanges, spot welded to a 12.70 mm (.050 in.) 4130 web sheet. The box beam extension arm substructure channels are brake formed and follow a constant spanwise taper. The box beam incorporates eight load fittings, four representative of the aft wing/fuselage attachment fittings and four outer corner load fittings for testing. The compression skin is .267 cm, 10 ply boron/ aluminum with $0^{\circ} \pm 45^{\circ}$, $0^{\circ} \pm 45^{\circ}$, 0° ply orientation. To reduce cost the tension skin is .254 cm (.1 in.) gage stainless steel since only the buckling capability of the B/Al compression skin is of interest. All box beam structural elements and skins are assembled with mechanical fasteners.

Instrumentation

The boron/aluminum wing box beam specimen was instrumented with axial strain gages and strain rosettes as diagrammed in Figure 16. The gages were positioned to monitor spanwise bending and shear stress distribution in both tension and compression skins, stress concentration around the bolt holes and initiation of buckling in the compression skin.

Loading

The box beam was loaded at the eight load fitting bolt holes to produce a condition of pure bending in the center section. This condition with total ultimate applied load of 38.6 kN approximates the critical 5g maneuver load condition. The box beam test set up is shown in Figure 17.

Test

After several initial load cycles to 30% D_kL_oL to exercise the specimen a run to failure was made. Buckling of the B/Al compression skin initiated at a load of 288.0 kN comparing well with analysis based on simply supported orthotropic plate theory which predicted initiation of buckling at a load of 314.1 kN. The early onset of buckling may be attributed to actual B/Al compression skin material properties being somewhat lower than those used in the analysis. The specimen continued to sustain increased loading after onset of buckling up to 612.9 kN, at which catastropic failure occurred. The failure is shown in Figure 18. The results of this test were used to substantiate the full-scale compression wing skin design for buckling stability.

WING SUBCOMPONENT

Design

In order to evaluate the behavior of the wing design in the area of highest tensile and compressive stresses, which is adjacent to the aft wing-to-fuselage attachment location, a second development test specimen was designed, fabricated, and tested. This was a subcomponent, outlined in Figure 19, which contained significant design details of the actual wing, with some minor alterations to simplify its fabrication and to provide test load application.

The tension and compression B/Al skins maintain constant ply thickness of 13 and 16 plies respectively over the entire subcomponent surface area. The ply orientation scheme of the skins is identical to that of the full-scale wing's center section. The boron-aluminum skins were fabricated by Amercom, Inc., including the countersunk holes which were made by electric discharge machining, Figure 20.

The substructure parts shown in Figure 21 which stabilize the skins at a constant depth of 4.10 cm were made and assembly operations performed at NAVAIRDEVCEN. At the subcomponent root end the wing center section skin splices are accurately represented by double row rivet attachment to .476 cm (3/16 in.) steel splice plates. These splice plates are supported

by a solid aluminum spacer bar which allows the complete assembly to be clamped for a cantilever test load set up. At the subcomponent free end, a 2.54 cm (1.0 in.) Al plate is fixed for test load application. The complete subcomponent assembly is pictured in Figure 22. The detail design drawings for the subcomponent are attached in the foldouts.

Test Loading and Instrumentation

Test loads to be applied to the B/Al wing subcomponent were determined with the aid of a NASTRAN loads analysis. This analysis resulted in a set of test loads which when applied to the subcomponent produced a stress field in the B/Al skins similar to the stress field present in the actual full scale wing skins when subjected to the 5g maneuver load condition.

The test set up shown in Figure 23 consists of the subcomponent mounted to a strongback testing facility; loads were applied to the specimen through two independent sets of wiffle trees by manually operated hydraulic jacks.

The subcomponent was instrumented with 73 strain gages and three deflection transducers. The gages monitor critically stressed regions on both tension and compression skins and are also paired internally and externally on the compression skin to check for initiation of buckling as shown in Figures 24 through 26.

The test load procedure was as follows:

- 1. Apply 30% D.L.L., 10% increments, check strain and deflection data.
- 2. Apply 50% D.L.L., 10% increments, check strain and deflection data, re-apply 50% D.L.L., 2 cycles.
- 3. Apply 100% D.L.L., 10% increments, check strain and deflection data, re-apply 100% D.L.L., 4 cycles.

Test Results

After initial loading to 30% D.L.L. strain and deflection data was plotted. Referring to Figures 27 and 28, typical strain and deflection vs. load plots from the test data reveal nonlinear, inelastic behavior exhibited by the B/Al skins. The second applied load cycle to 50% D.L.L. yielded approximately linear elastic response in the skins up to the previously applied load level (30% D.L.L.). Subsequent loading above the 30% D.L.L. level resulted in a continuation of the nonlinear inelastic behavior in the skins. Additional load cycles to the 50% D.L.L. level yielded repeatable linear elastic response in the skins.

The initial run to 100% D.L.L. resulted in a failure at the 70% D.L.L. level. Again nonlinear inelastic behavior was exhibited by the skins once the previously high loading point was exceeded (50% D.L.L.). The failure occurred in the tension skin, a crack initiating at the corner radius, just outboard of the aft bolt hole, and propagating across the skin following a

path of minimum net section (see Figure 29).

This failure can be attributed to stress concentrations present at the corner radius which are amplified by the close proximity of a fastener. Strain levels monitored on both tension and compression skins at time of failure were similar to those predicted by analysis except in the local failure area. In addition, the load-strain and load-strain and load-deflection behavior of the specimen was highly non-linear, and large permanent deformations were present after testing at various load levels under the failure load.

Stress strain behavior of a tensile coupon cut from the same laminate as the B/Al subcomponent tensile skin is shown in Figure 30. Stress/strain data for the 6061 Al matrix is also plotted. The early onset of plasticity in the Al matrix appears to have a significant influence on the overall stress/strain response of the B/Al composite when subjected to loading. The B/Al laminate begins exhibiting inelastic behavior at approximately the same strain level that the 6061 Al becomes plastic.

FINAL WING DESIGN CRITERIA

Based on the results of this test, a review of stress-strain behavior of tensile specimens and some limited data on stress concentration in drilled holes, the following critieria was formulated for final design of the wing skins:

Nominal limit load stress ≤ 360 MPa Strains at limit load ≤ 2000 Mm/m Stress concentration factor = 1.5

It was the above design criteria which dictated the need for additional B/Al ply build up to 24 plies, in the aft attachment region, on both tension and compression wing skins to relieve stress concentrations due to attachment holes.

Final analysis using the NASTRAN finite element program was performed to confirm the stress and strain levels in the wing. The estimated total weight is 52.8 kg, 30 percent less than that of the production wing, which was designed for only 422 degrees K. Of the total weight, the skins comprise 26.3 kg, or 50 percent. The leading edge, substructure, centerline splice, and rivets and fittings weigh 5.9, 28.3, 3.2 and 4.5 kg respectively.

CONCLUSIONS

In this program, a design has been developed using metal-matrix composites to achieve high temperature capability and reduced weight. Much has been learned about the behavior of boron-aluminum and critieria for its use in aircraft structures. Additional development work would be required before it could be incorporated into an actual system. In particular, more data is

needed on fatigue and on stress concentrations in loaded holes both at low and high temperatures; basic fracture characterization should be performed; laminate tailoring should be investigated to minimize these effects as well as those due to the non-linear behavior and permanent deformations.

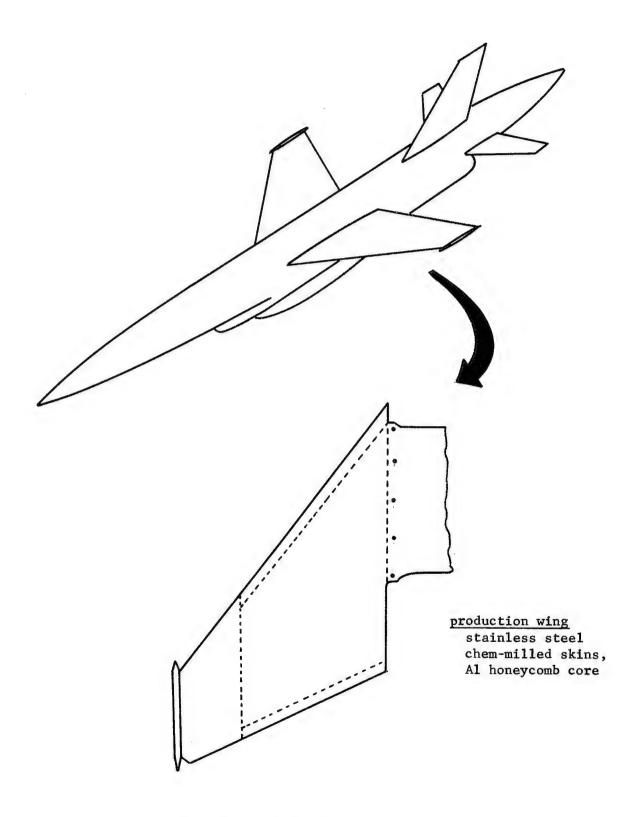


FIGURE 1 - BQM-34E RPV

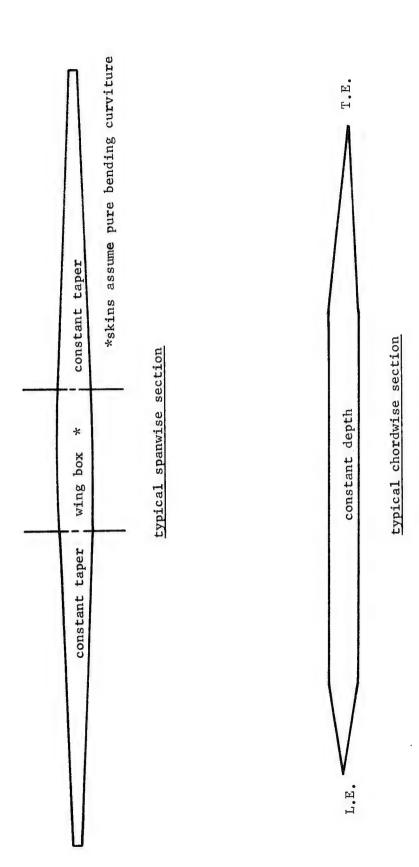
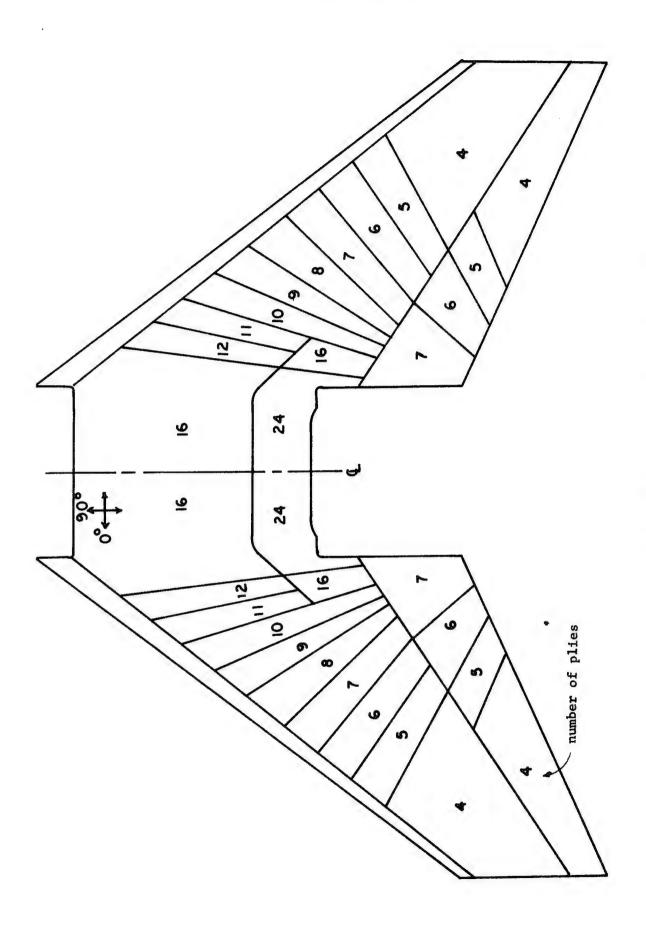
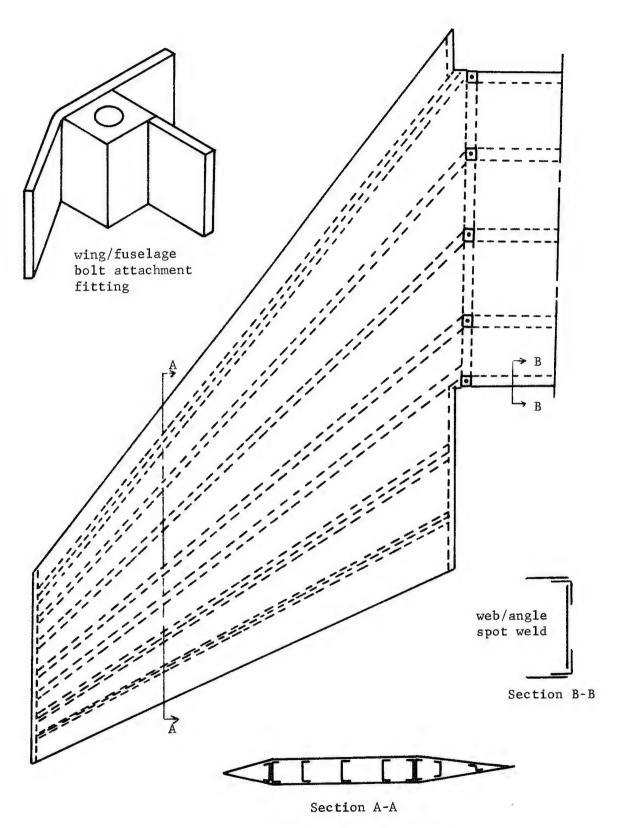


FIGURE 2 - B/AI WING SECTION GEOMETRY

FIGURE 3 - B/AI WING BASIC SKIN CONFIGURATION

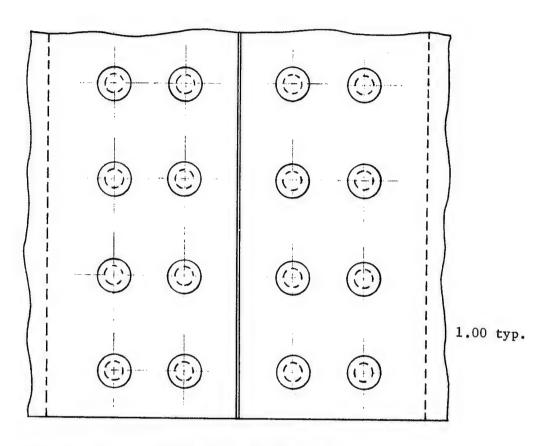
FIGURE 4 - B/AI WING TENSION SKIN LAMINATE DESIGN

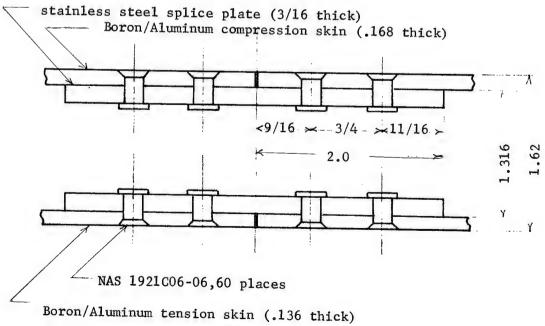


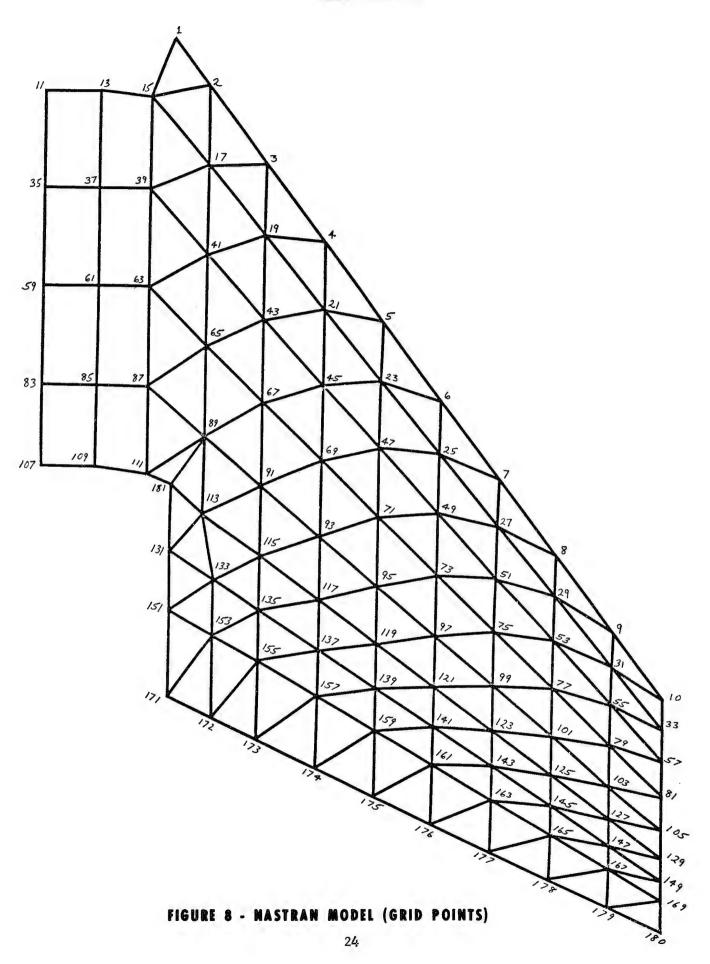


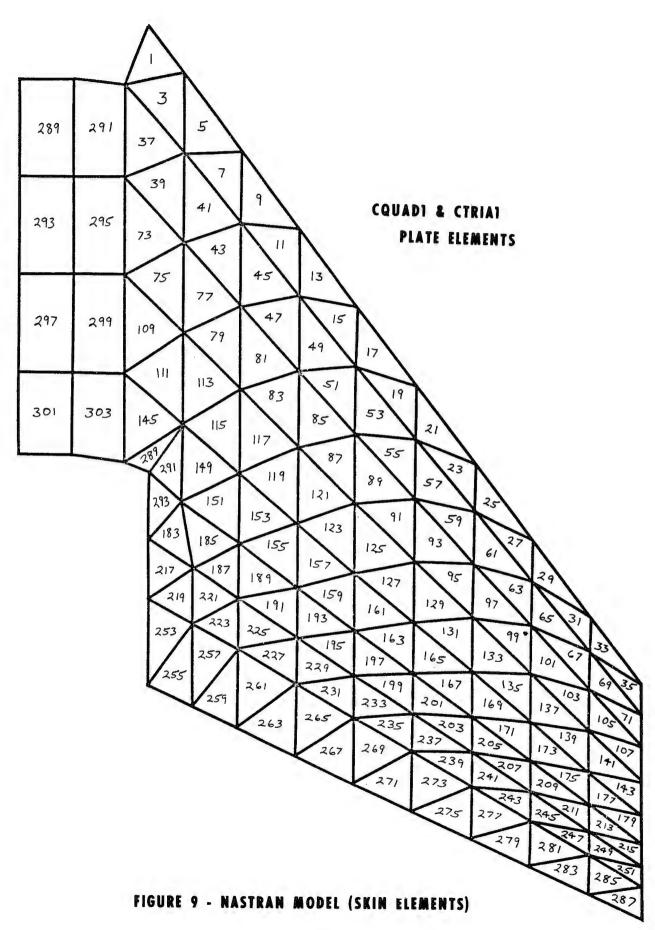
Typical Chordwise Section

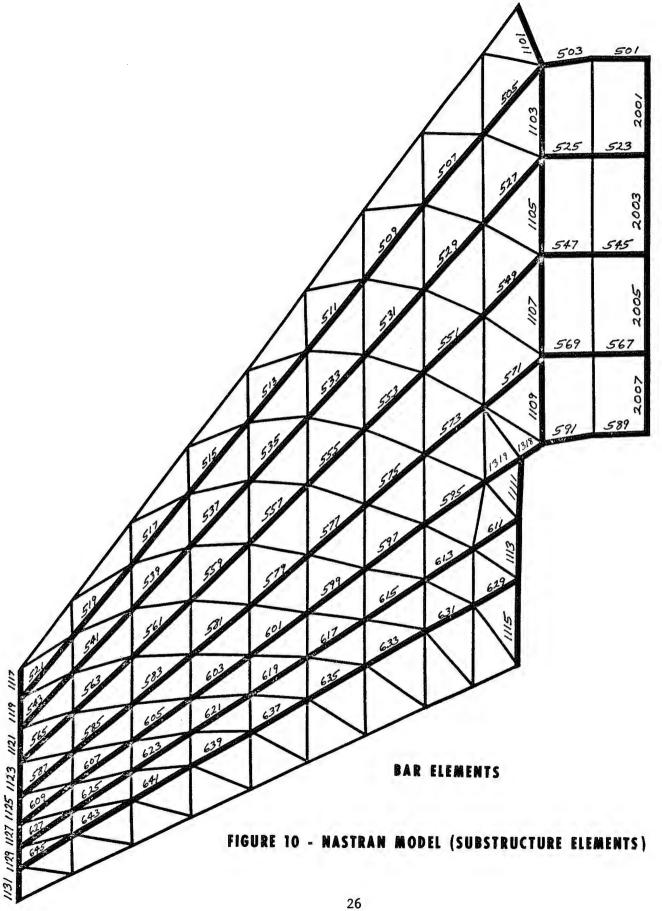
FIGURE 6 - B/AI WING SUBSTRUCTURE

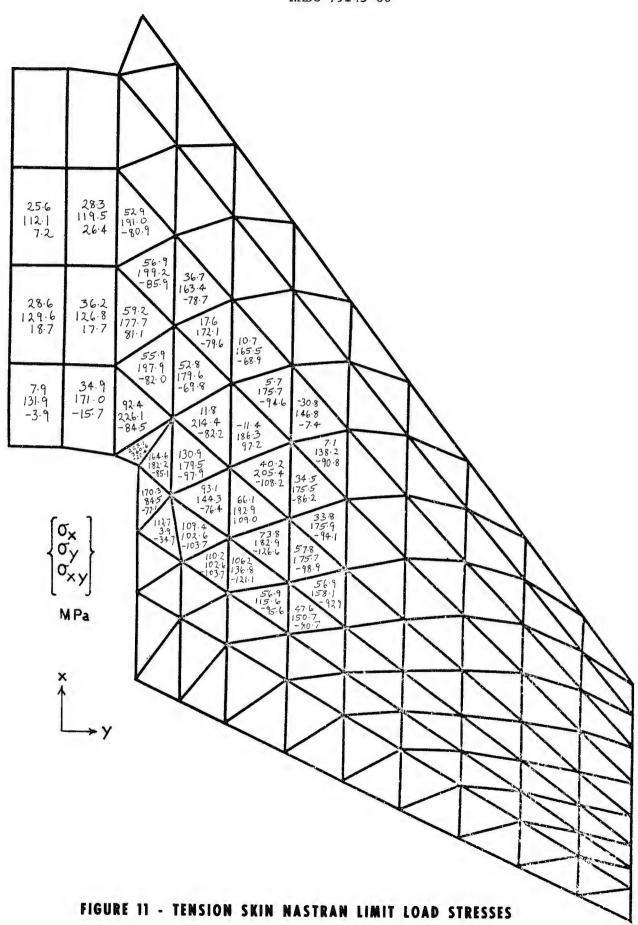


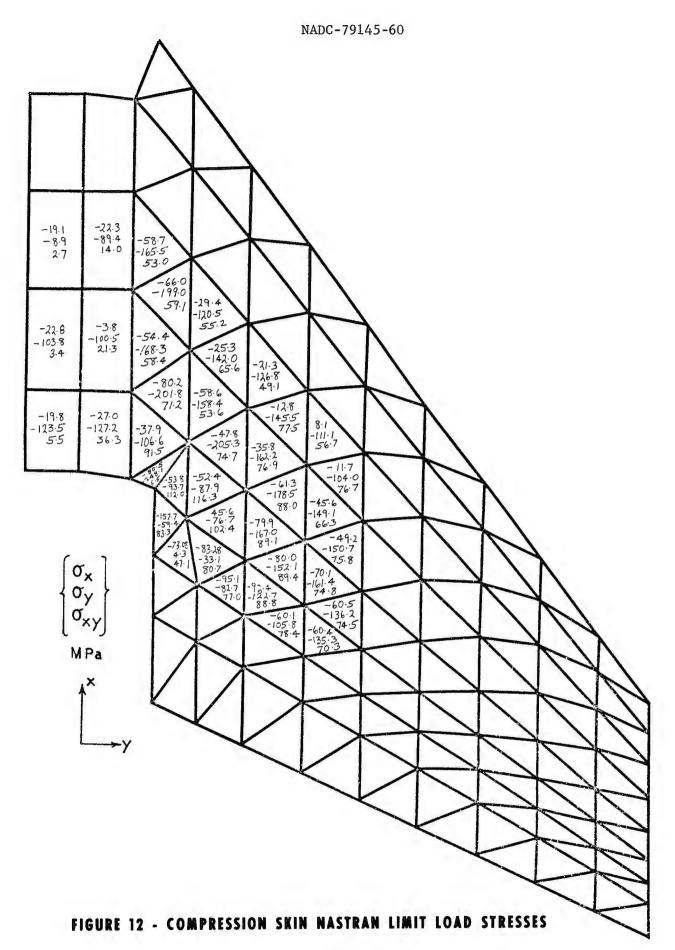


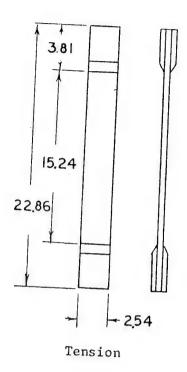


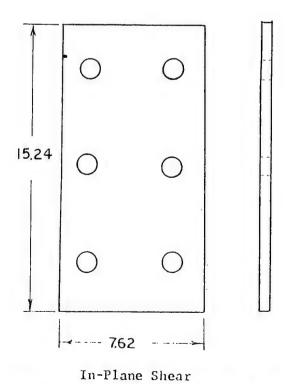












IGURE 13 - MATERIAL COUPON SPECIMENS

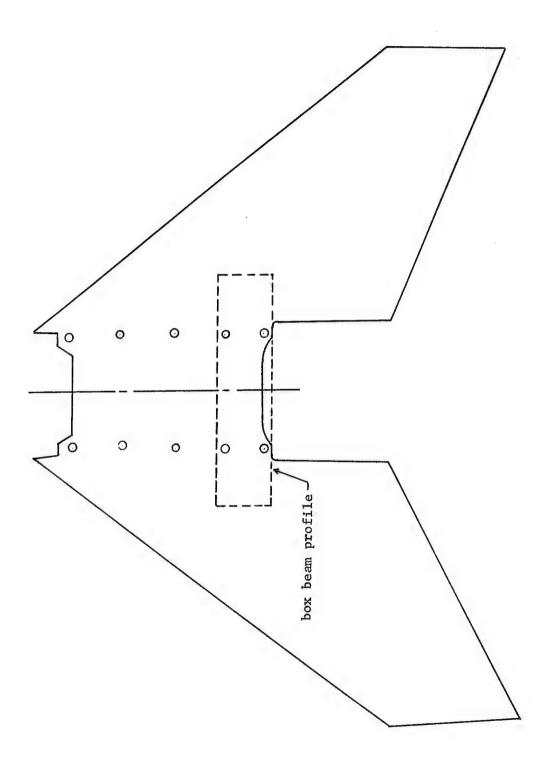
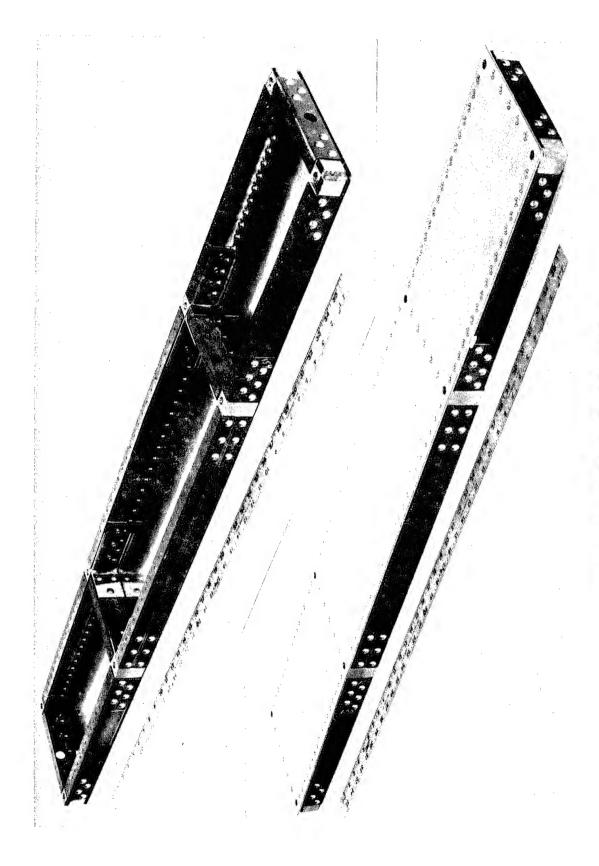
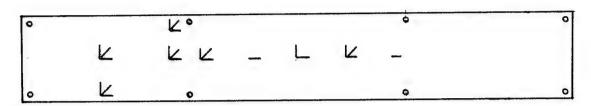
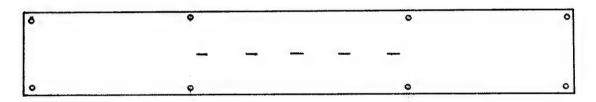


FIGURE 14 - BOX BEAM SUBCOMPONENT PROFILE

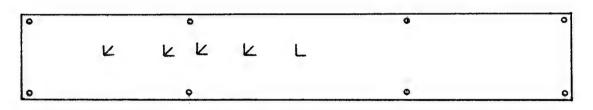




compression skin - outer surface



compression skin - inner surface



tension skin - outer surface

Key

- axial gage
- 0 / 90 gage

- rosette

Figure 16 - Box Beam Subcomponent Instrumentation

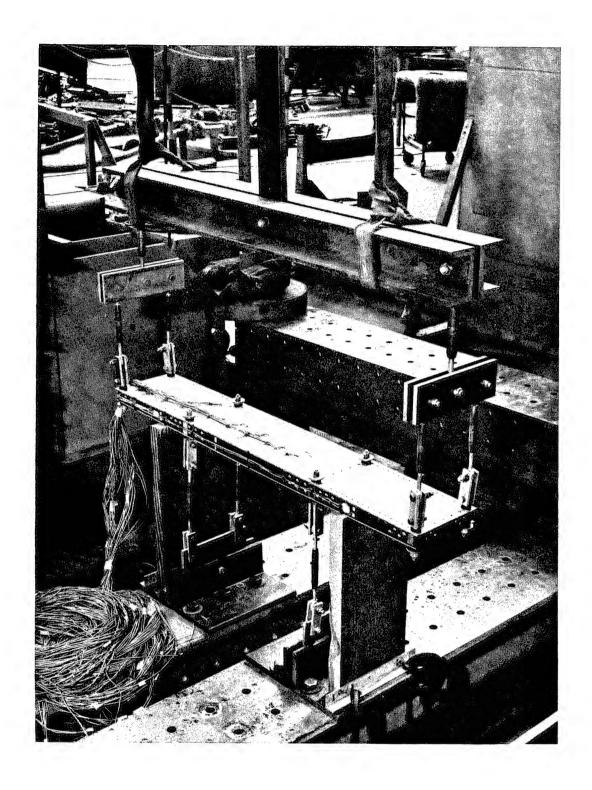
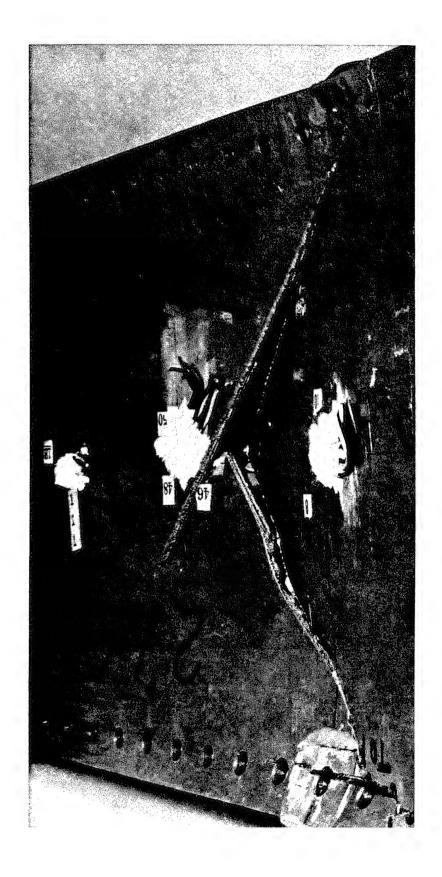
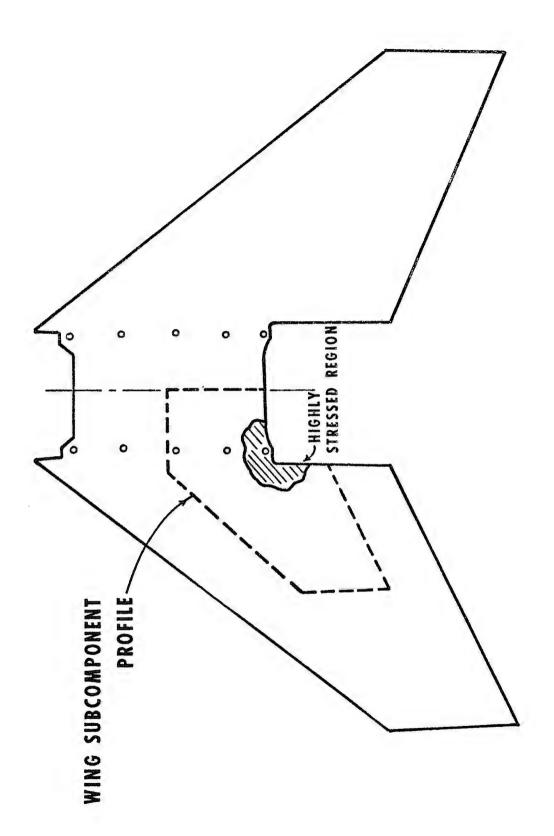
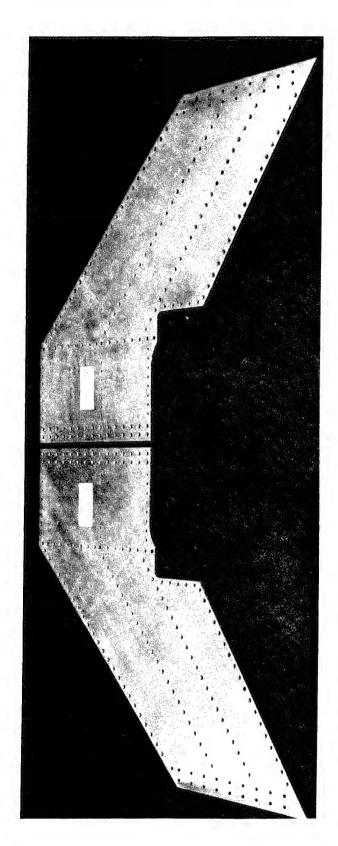


FIGURE 17 - BOX BEAM SUBCOMPONENT TEST SETUP



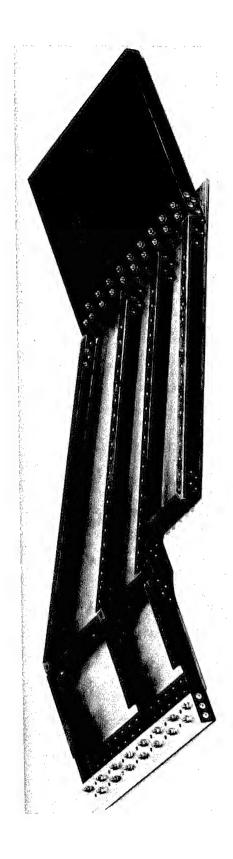




COMPRESSION SKIN - 16 PLIES

FIGURE 20 - B/A! WING SUBCOMPONENT SKINS

TENSION SKIN - 13 PLIES



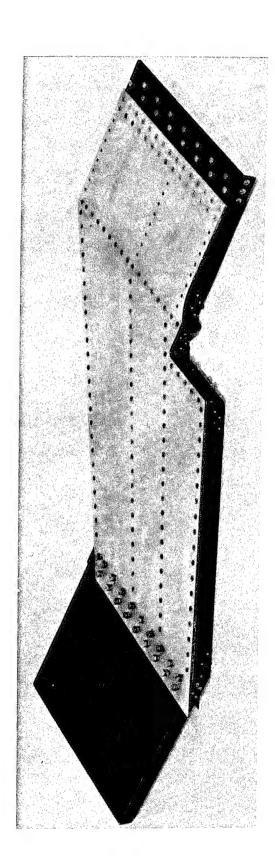
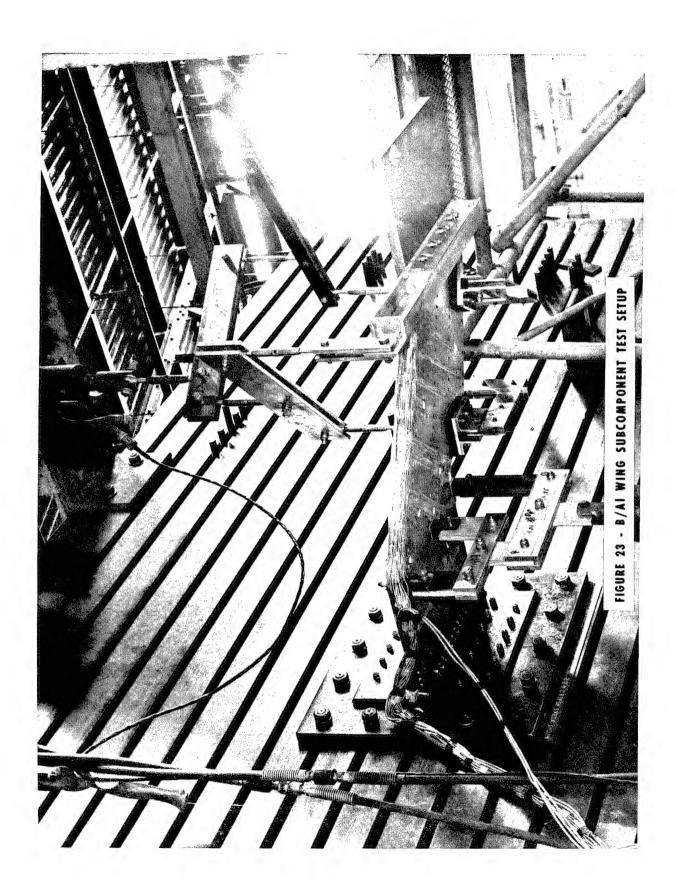
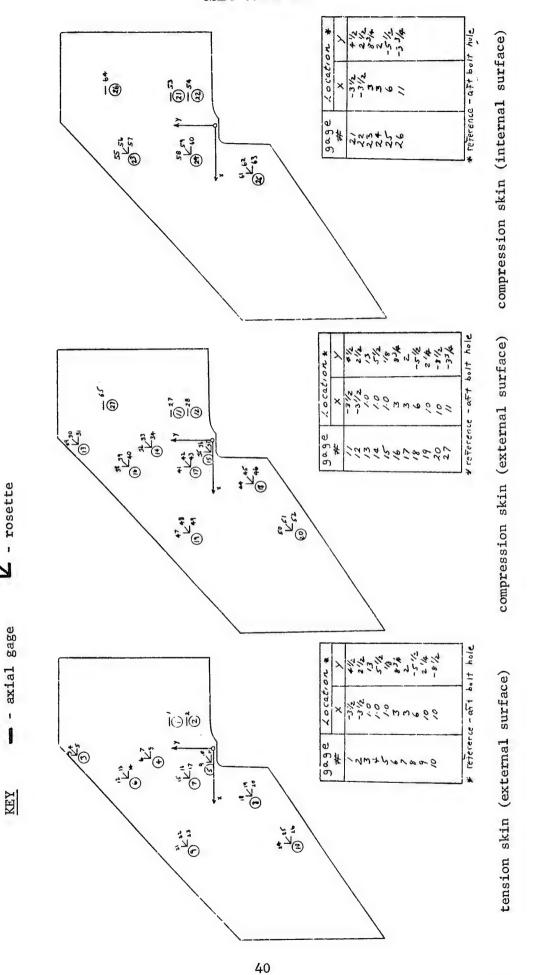


FIGURE 22 - W B/AI WING SUBCOMPONENT FINAL ASSEMBLY

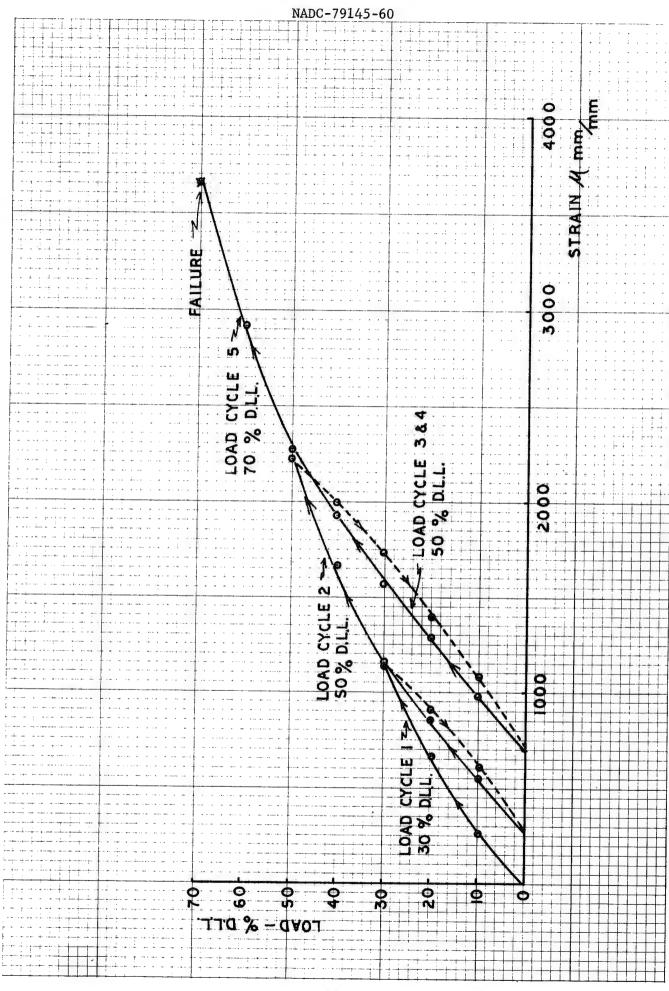


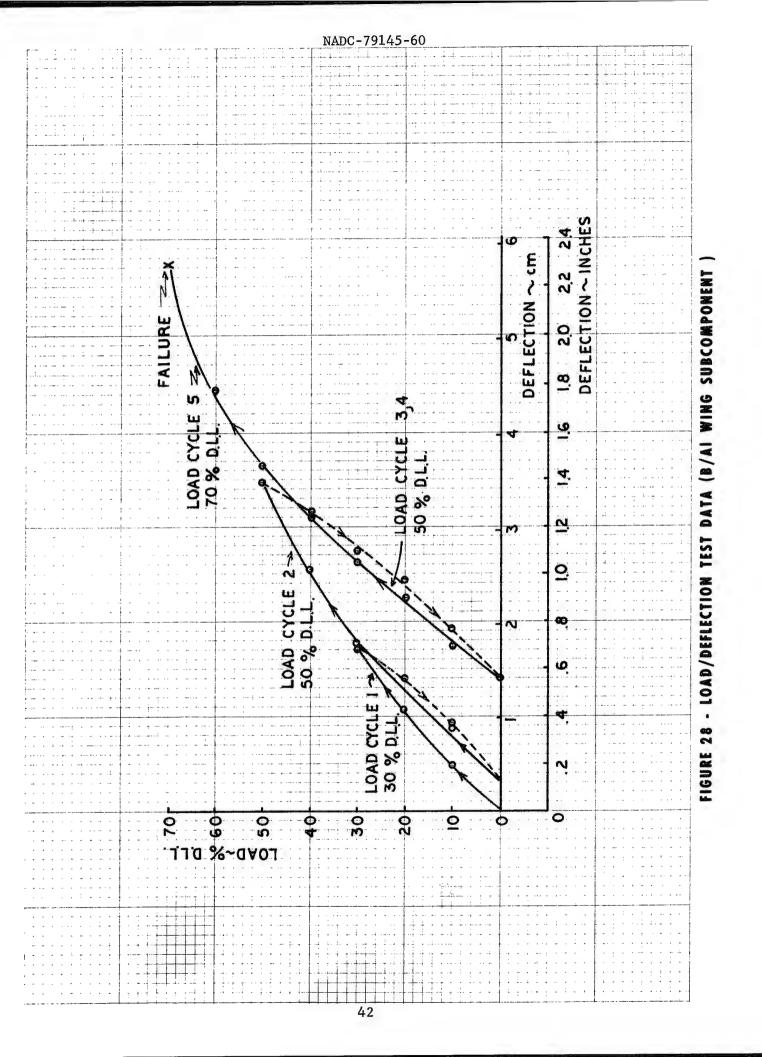


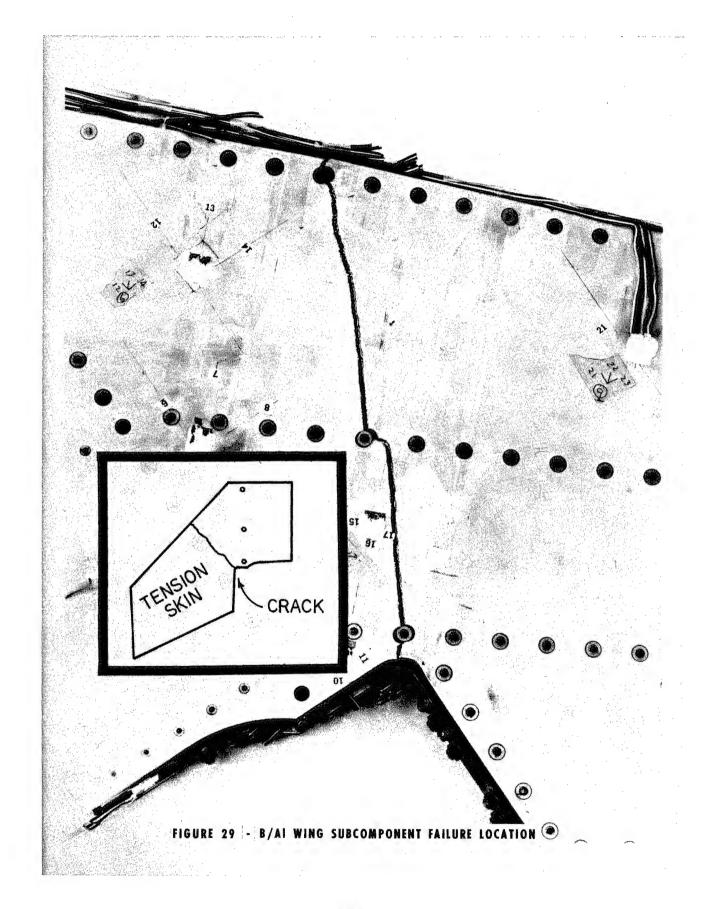
- rosette

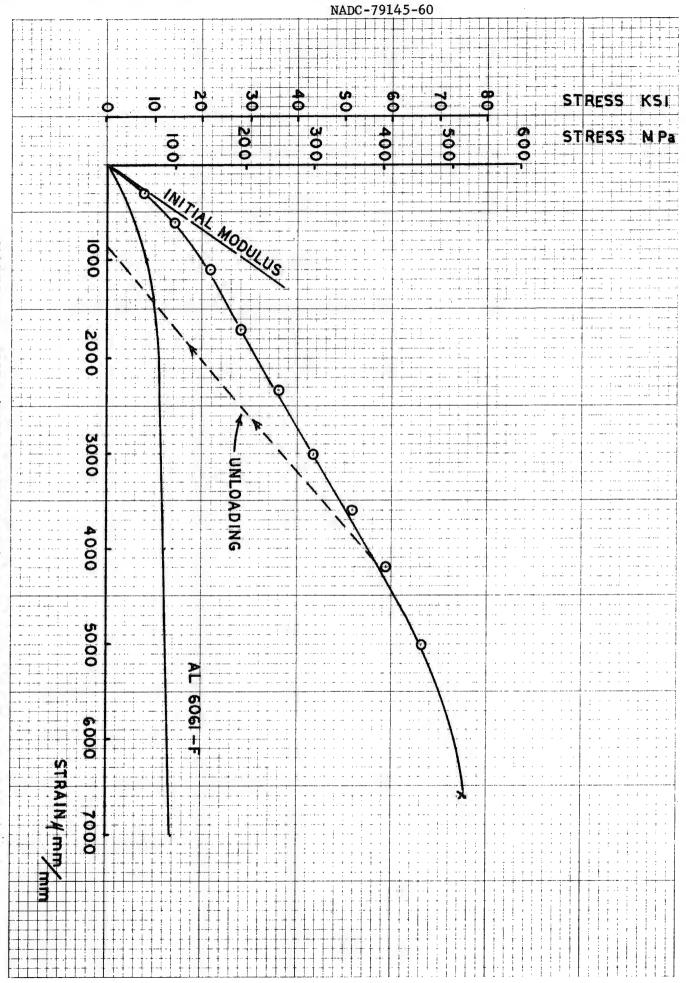
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Figures 24,25 & 26 - B/Al Wing Subcomponent Instrumentation









NASTRAN Element Number	Nxult KN/m	Nxyult KN/m	Nxcr KN/m	Nxycr KN/m	<u>Nxult</u> Nxcr	Nxyult Nxyer	Margin of Safety
290	-1985	15.6	-9619	13518	.205	.001	3.84
292	-750	896	-7996	13518	.094	.066	6.81
294	-1261	666	-1737	17764	.746	.037	.37
150	-469	620	-3055	5263	.154	.118	3.60
116	-822	299	-800	1072	1.028	.279	09
154	-501	267	-698	1091	.718	. 245	.26
156	-409	547	-4619	6827	.089	.080	6.36
158	-377	175	-425	641	.887	.273	.04
190	-327	237	-406	808	.805	.293	.11
194	-271	140	-265	403	1.023	•347	11
162	-284	110	-294	428	.966	.257	03
198	-223	88	-251	344	.888	.256	.05
304	-1020	291	-4588	6094	. 223	.048	3.31
166	-216	66	-211	289	1.024	.228	07

Table 1 - B/Al Wing Compression Skin Critical Buckling Loads

Laminate Type	# Specimens	Ult. Tensile Stress (MPa)
0/ <u>+</u> 45	5	492.7
90/ <u>+</u> 45	5	189.3
<u>+</u> 45	6	327.3

Table 2 - Results Tensile Coupon Tests

Laminate Type	# Specimens	Ult. Shear Stress (MPa)
0/ <u>+</u> 45	5	257.4
0	5	131.4
<u>+</u> 45	5	309.6

Table 3 - Results Rail Shear Coupon Tests

Laminate Type	E ₁ (GPa)	E ₂ (GPa)	G (GPa)	12	21
0/ <u>+</u> 45	158.2	131.3	50.5	.331	.307
<u>+</u> 45	137.2	137.2	54.9	.364	.364

Table 4 - Experimental Material Property Constants

Fibers in Test Direction (%)	23.1 23.1 30.1 30.1 18.8 18.8 31.3
Fil Tes tic	
Failure Strain m/m	.0080 .0075 .0066 .0066 .0077 .0065
Ult. Stress MPa	541.9 530.9 532.3 612.3 484.7 450.2 528.8 504.0
Ult. Load KN	23.8 23.5 23.9 27.9 26.5 24.5 29.0 27.2
Prop. Limit MPa	88.3 86.2 97.9 95.2 53.8 51.0 75.8
$ m E_1$ GPa	175.8 173.1 120.0 157.2 151.0 134.5 163.4
X-Sect Area (cm ²)	.445 .447 .462 .554 .551 .557
Specimen Number	6240P-A1 6240P-A2 6240P-A3 6240P-A4 6241P-A1 6241P-A3 6241P-A3
Test Dir.	06 0 0 0 0 0
Wing Skin	7777 7777

(Tension) ----(0,90,+45,-45,0,-45,90,+45,0,+45,-45,90,0) 13 ply H Wing Skin

(Compression)----(0,90,+45,-45,0,-45,+45,0,0,+45,-45,90,-45,+45,90,0) 16 ply Wing Skin -2

APPENDIX A

NASTRAN BULK DATA

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VASTRAN

A-2

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ND VEMBER	0 E C K E C H G 54,83,84,87,88,107,10 70P.
IALY3IS• EXP• PROP• II-78 + NEW GII DF ELEM 289+297(11-2-78)	SET 5= 11,12,15,16,35,36,39,40,59,50,63,54,93,84,87,88,107,108,111,112 SPCFORCES = 5 ELFORCE = ALL DISPLACEMENT = ALL TITLE = 8/AL WING STAFIC ANALYSIS, EXP. PROP. SUBTITLE = SKIN CHANGES OF 10-31-78 + NEW GII OF ELEM 289+290(11-2-78) SPC = 10 SPC = 10 SPC = 10
B/AL WING STATIC ANALYSIS, SKIN CHANGES OF 10-31-78 +	CARD COUNT

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STATIC	S OF
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BIAL AING	SKIN

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- (PAR 524 .6892 .7500 .0000 .7500 .6959 .0000	P2R 52
2 0	P23 524 660026 359127 0.	
0 4	043 524 - 4622 11 • 4059-1 • 9930-2 • 2597-2 • 1570-4 0.0	
. 70	P28 525 - 651013 - 348648 0.	28 82
9	3AR 525 11 .4069-1 .9930-2 .2507-2 1570-	
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. 8 . 7 . 8	972-2 .1218-4 0.0 .6420 .0000 .6241	972-2 .1187-4 0.0 6241 .00005746	972-2 .1137-4 0.0	.6241 .0000 .5746	972-2 -1148-4 0.0	972-2 :1148-4 0.0	.5745 .0000	972-	5357 -0000	972-2 .1113-4 0.0	357 .00	972-2 .1073-4 0.0	.4968 .0000	972-2 .1073-4 0.0	0	972-2 .1029-4 0.0	.4473 .0000	972-2 .1028-4 0.0	.4473	971-	3979 .0000	971-2	.3979 .0000	971-2 .0935-4 0.	3484 .0000	971-2 .093	.3484 .0000	971-2 .0983-4 0.0	.2335 .00	1971-2 .0883-4 0.0
	1 1.0792-2.3	1 .9554-2 .3	-9554-2	٠	1 .8029-2 .3	0.00°	0 .7500	- 1	0 . 7500	1 .6818-2	00 .75	.5587-2	0 .75	1 .5587-2	475	1 .4356-2	62 . 00	1 .4355-2 .3	00 . 7500	1 .3312-	00 - 1500	.3312-	00 4.4 0	1 .2342-	000 .7500	1 .2342-	00 - 7500	1 -1508-2	00 .75	1 .1503-2 .3
1 2	528 11 .4059-	529 .467863 .554310 0. 529 11 .3958- 5295993 .7500 .00	529 .454276 .568453 0. 530 11 .3958-	530 .59937500 .00 530 .454276 .568463 0.	531 11 ±3825-	531 -435365 -588161 0.	5325552 750000	532 .435365 .583161 0. 533 11 .3709-	533 5162 7500 .03	534 11 .3709-	5345162 7500 00 534 .417581 606686 0	535 11 .3576-	5354720750000	536 11 .3576-	536 7 44720 77550 .00	550 *542442 *664117 U* 537 11 *3428-	5374226 -7500 -03	538 11 .339	5384226 .7500 .00	539 11 .327 .327 .327	53937327500 .0	539 •341361 •686082 0• 540 11 •3279	540 .3732 .7500 .	540 .341361 .686082 0. 541 11 .3115-	5413185 .7500 .03	541 .3066// . (22212 0. 542 11 .3115-	542 .31857500 .00	542 .306677 .722212 0. 543 11 .2943-	543 2609 - 7500 - 00	544 11 .2943-
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23	P23 551	454276 .56846	0						
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25	PAR 55	5993	.00	.7500	.6241	0000	.5746	.0000+P2R	552
26	P23 552	454276 .56846	0						
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29	P23 553	435365 .58816	•						
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35	P23 555	417581 .60668	0						
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37	J.	5162 .75		500	.5357		.4968	.0000+P2R	556
33	P28 556	581 .60668	0						
33	3A2 5		.3576-1 .	7	97	073-4 0.0		Va	10
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4456- 4554- 45	-56234136168608 553 11 5633185 .750	484E. 0000. 679E. 0077. C000. 0	2R 56
667- 668-	553 11 5633185 .750	• 0	
458- 460- 460- 460- 460- 460- 400-	563 3185 .750	1 *2342-2 *3971-2 *0935-4 0.0	AR 5
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4682- 4683- 4644- 4655- 4765- 4776- 47	554 .3185 .750	.0000 .7500 .34	28 56
463- PBA 464- +PA 455- +PA 465- PBA 465- PBA	. 554 -: 306677 : 7222	20.	
464- +PA 465- +P2 466- P3A 467- +P3	555 11	.2943-1 .1508-2 .3971-2 .088	+PAR 565
465	565 2609 .750	.0000 .75002885 .	2R 56
465- P3A	-55525600376457	2-0-5	
467-	556 11	.2943-1 .1508-2 .3971-2 .0883-4 0.0	+PAR 56
	A 566 .2609 .7500	0 .0000 .7500 .2885 .0000 .2334	.0000+P2R 566
468+22	-566256008 -76		e entre e transfer en entre en entre en entre en entre en en en en en en en en e
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d 425-	559 11	.6043-1 1.4506-2.3744-2 .5227-4 0.	3 56
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478- PBA	570	.6043-1 1.4506-2.3744-2 .5227-	AR 57
Vd+ -625	570 .6622 .7	.0000 .7500 .6825 .0000	0000+P2R 570
480	570 657476	• 6	!
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489+	. 454276 . 56846	· O	
490- P34	574 11	.3958-1 .9554-2 .3972-2 .1187-4 0.0	+PAR 5
491- +PA	574 .5993 .750	9. 0067. 0000. 0	28
492+72	574 .454276 .56846	3 0.	
493- PBA	575 1	25-1 .8029-2 .3972-2 .1148-	IR 5
Vd+ -565	575 5552 .750	0000 . 7500	
495 + 52	-575 .435365 .58816	1 0.	
496-	576 11	.3825-1 .8029-2 .3972-2 .1148-4 0.0	+PAR 57
Va+ -265	52 .750	0000 .7500 .5745 .0000 .5357	.0000+P2R 57
498+P2	576 .435365 .53816	1.0	
4 66-	8 577 11	.3709-1 .6818-2 .3972-2 .1113-4 0.0	AR 5
+ -002	5775162 .7500	0000 .75095357 .00004968	

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DATA ECHO	6 7 8	3972-2 .1113-4 0.0	357	3972-2 .1073-4 0.0	0000 • 0000	3972-2 .107	. 80	3972-2 .1028-4 0.0	. 0000 • £25	3972-2 .102	. 4473 .0000	3971-	3979 .00003484	3971-2 .0984-4 0.0	.3979 .0000	0 7-35-00 6	43434 ·0000		-2 .0935-4 0.0	*3484 *0000 *2885	1-2 .0893-4 0.	2885 -00002334	3971-2 .0883-	.2885 .0000 .2334	2 -5421-4 0-	6959 .0000	3745-2 -5421-4 0.0	.6959 .0000	0 / 5000 6 7750	0.0 4-1226. 2-4	740.	-2 .5227-4 0.0	55 • 00	72-2 .1213-4 0.0	20 . 0000	.3972-2 .1218-4 0.0
S D R T E D B U L K	3 606686 0	.3709-1 .6818-2	605685 0.	.3576-1 .5537-2	629175 0.	.3576-1 .5587-2	•7500 •0000 •75	63	.7500 .0000 .75	.3428-1 .435	.75	.3279-1 .3312	. 7500	.3279-1 .3312-	0067. 0000 0087.	686082 0. -3115-1 -2342-	7	722212 0.	.3115-1 .2342-2	72221.2 0.	.2943-1 .1508-2	.7500 .0300 .75 764575 0.	.2943-1 .1508	27500 .0000 .75	164272 0.	0057. 0000. 0057.	362600 0.	.7500 .0000 .7500	362600 0.	2-506+T T-5508		.5043-1 1.4505	372309 .3000 .750 372309 0.	.4059-1 1.0792-2	50	.4059-1 1.0792-2
	73 577	578 1	23 578	AR 579 1	23 579 .395992	AR 530	A3 580 . 44720 . 72 580 . 345442	AR 591	A3 581 4226 28 581 -369858	AR 582	+PA3 582 4226 +P28 582 . 369858 .	A2 533	A3. 583 3732 22 583 341261	AR 584	AR 584 :373	2	AR 585 3185	28 585 .30667	AR 585	28 536 .306677	AR 587	+PA3 587 2609 +P23 587 -266008 -	AR 583 1	A2 588 .2609	AR 589	AR 589 68	28 239 .65640 AR 530	43 590 .68	28 593 •66640 As	43 591 6622	23 591 .	A2 592	28 592 •657476	AR 543	22 593	A2 594
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554	+PAR 59	iv a	.7500	.0000	. 7500	6241	0000	5851	.0000+P2R	265
555	BAR	596	7 7 7 7	3974-1	9735-	-2	1	0	⋖	5
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565	BAR	539	6000	• 0	7133-	-2	1		<	50
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482	PAR 60	.378		0000	. 7500	4084	.0000	3484	.0000+P28	609
586	BAR OU	5 • 344209 606 1	Z C C	3295-1	3426-2	,	0 4-6850	c	0	4
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593	PAR 50	.3185	.750	0000	.7500	3484	0000		82840000	809
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595	BAR	609			15.08-2.	71-2 .	1	0	+PAR	609
595	PAR 60	3 2603	. 764575	0	5		0000.	2334	2	9
593	SAR	510		43	-2	-2	0883-4 0.		4	6
599	PAR 51			.0000	.7500	.2885	.0000	.2334	.0000+P2R	
900	P2R 61	*25500	57	• 0	•					

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5 C	34 13 2.0891-1.1567-4 8.44235 .5267-4 0.0	+PAR 7
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33.2	PAR 742 .0000 .0000 .0150 .0000 .0000 .0000	.0000+P2R 742
334	742 .8333 1.0060 0.	
33. 10.	343 743 13 .9829-1 .0737-4 3.51591 .2943-	PAR 7
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+ P2	13210	000062 00	.00006420	.0000+P2R1101
P3A	01 .308358 .7204	0.		
	1102 11 .3	146-1 .2937-2 .3971-2 .09	0.0 4-4	48110
V d+ +b V	023210 .	. 1500 • 0000.	0	RII
- +P2		0.		
- P3A	11	960-1 2.2503-21.5454-2.66	33-4 0.0	0110
A 4+	03 76420 1.0000	.0300 1.0000 6423	0000	P2811
20+	3267			944
PBA	1104 11	960-1 2.2503-21.5454-2.66	33-4 0-0	401104
V d+	5420 1.0000	.0000 1.0000 .6420	0000	
- +P2	04 .403267 .			-
P3A	1105 11	960-1 2.2503-21.5454-2.66	33-4 0.0	0110
- +PAR1	105 6420 1.0000	00000 1.00006420	0000	2811
- +P231	3267 .			
- PBA2	11 00 11	960-1 2.2503-21.5454-2.66	33-4 0.0	+PAR1106
Vd+	06 .6420 1.000	.0000 1.0000 .6420	0000	28110
- +P2	06 .403267 .			
P B A	1107 11	960-1 2.2503-21.5454-2.66	33-4 0.0	R110
V d+	76420 I.0000	0 1.00006420	0000	2811
- +P2	07 .403267 .			
- P8A	11 08 11 .7	960-1 2.2503-21.5454-2.66	3-4 0	AR110
V d+	08 6420 1.0000	0000 1.0000 .6420	000	.0000 +P2R1108
- +P2	3 .40326	0.		
- P3A	11	960-1 2.2503-21.5454-2.66	33-4 0.0	AR110
V d+	9 6420 1	.0000 6420	0000	.0000+P2R1109
- +P2	. 403267 . 62	.0		
- PBA		950-1 2.2503-21.5454-2.66	- 1	+PAR1110
V d++bV	1110 . 6420 . 1,0000	0 1.0000 .6420	.0000 .6420	22111
- + P2	.403267	0.		
- P8A	1111 11 .2	785-1 .8150-2 .2702-2 .03	71-4 0.0	+PAR1111
V d+	•	.0000 .75006420	0000 6840	28111
- +P2	•4759			
P84	1112 11	785-1 .8150-2 .2702-2 .03	71-4 0.0	2111
V d+	6630	.0000 .7500 .6420	.0000 .6840	.0000+P2R1112 -
- +P2	.475952 .538405	•0		
- PBA	1113 11 .2	545-1 .5602-2 .2702-2 .03	39-4 0.0	+PAR1113

	9 10 . .0000+P2R1113	.0000+P2R1114	. +PARII15	.0000+P2R1116	.0000+PAR1117	.0000+P2R1118	+PARII19	+PAR1120	+PAR1121.	.0000+P2R1122	.0000+P2R1123	+PAR1124	+PAR1125	+PAR1126	+PAR1127	+PAR1128	*PAR1129
DATA ECHO	6 7 8	2-2 .0339-4 0.0 . 4230 6840 .0000 .4230	2-2 .0250-4 0.0 4230 .00000000	02-2 .0250-4 0.0 .4230 .0000 .0000	2-2 .0226-4 0.0	2-2 .0226-4 0.0 .0000 0000 .2334	2-2 .0257-4 0.0 2334 .00002334	2-2 .0257-4 0.0 2334 .0000 .2334	2-2 .0257-4 0.0 2334 .00002334	2-2 .0257-4 0.0 2334 .0000 .2334	2-2 .0257-4 0.0 2334 .00002334	2-2 .0257-4 0(2334 .0000 .2334	2-2 .0257-4 0.0 2334 .00002334	2-2 .0257-4 0.0 2334 .0000 .2334	2-2 .0257-4 0.0 2334 .00002334	2-2 .0257-4 0.0 2334 .0000 .2334	2-2 .0242-4 0.0 2334 .00301232
RTED BULK	00 .0300 .7500	.2545-1' .5602-2 .270 .0000 .7500 .	.1877-1 .0727-2 .270 0 .0000 .7500	1 0. •1877-1 •3727-2 •27 • •0000 •7500 •	.1693-1 .0204-2 .270 0 .0000 .7500	.1693-1 .0234-2 .270 .0303 .7500 .	.1927-1 .0814-2 .270 .0900 .7500	.1927-1 .0814-2 .270 .0000 .7500 .	.1927-1 .0314-2 .270 .0000 .7500	.1927-1 .0814-2 .270 .0300 .7500 .	.1927-1 .0814-2 .270 .0000 .7500	.1927-1 .0814-2 .270 .0300 .7500 .	.1927-1 .0814-2 .270 .0000 .7500	.1927-1 .0814-2 .270 .0000 .7500 .	.1927-1 .0314-2 .270 .0300 .7500	.1927-1 .0314-2 .270 .0300 .7500 .	.1217-1 .0507-2 .270 .0000 .7500
S O	1	AR 1114 11 ARIII4 • 5535 • 75	A21115 2115 -75	4642 1 2115	601110 •264042 • 793 AR 1117 11 AR1117 •1167 •7	A3 1118 11 A31113 •1167 •75	42 1119 11 42 1119 11 5211192334 -75	AR 1120 11 AR1120 •2334 •75	0.04	343 1122 11 PARTILE 1795 PRETITE . 2334 . 75	AR 1123 11 AR1123 2334 - 75 ZZ1123 - 242241 - 7785	A21124 11 A21124 -2334 -75 221124 -242241 -7735	2 1125 11 011252334 -75 21125 -242241 -7785	A21126 .2334 .75	A2 1127 11 A21127 2334 -75 231127 -242241 -7735	A2 1123 11 A21129 .2334 .75 271129 .242241 .7785	48 1129 1129 1793 755 221129 196295 8255
C 4	2001- 2002-	003-	-200	0009	012	015-	013-	021-	024- 025- 026-	023- 023-	030-	0333-	034	039	042	040 040 047	0.40 0.40 0.50

NOVEMBER 2, 1978	
BIAL WING STATIC ANALYSIS, EXP. PROP.	SKIN CHANGES OF 10-31-78 + NEW GII OF ELEM 289+290(11-2-78)

NASTRAN 12/16/77

4		
COUNT	7 2 4 5 6 7 .	9 10 .
051	BAR 1130 11 .1817-1 .0507-2 .2702-2 .0242-4 0.0	+PAR1130
0	30 .1783 .7500 .0000 .7500 .2334	.0000+P2R1130
053	P2R1130 .196295 .825725 0.	
7 7 7	843 1131 11 .1583-1 .0051-2 .2702-2 .0211-4 0.0	+PAR113
200	FAXIL31 U615 . 7500 . U000 . 7500 1232 . 0000	.0000+P2R1131
יו טרי	PC41131 *U/7829 *94/435 U.	
1 0	0.0 1.13C 11. 0.001 - 1.0001 - 1.0001 - 1.0001 - 0.0011 -	+PAKIL
500	P221132 077829 067436 -: 0.	*0000+FZK113Z
090	8AR 1301 13 1-0943-1-0821-4 4-85	+PAR1301
190	PAR1301 .0000 .0000 .0150 .0000 .0000 .0000	PZRI
062	P221301"" - 8333"" 1:0000 ""0.	
90	BAR 1302 13 2.0	+PAR1302
90	PAR1302 .0000 .0000 .0250 .0000 .0000 .0	813
5	P2213322	
990	BAR 1303 13 4.1553-1.8656-4 23.9	+PAR1303
290	PAR1303 .0000 .0000 .0250 .0000 .0000 .0	8130
963	P2R130	
690	BAR 1304 13 4.1	+PAR1304
070	PA31304 .0000 .0000 .0250 .0000 .000	3
110	P2R1304 8333 1.0000 0.	
272	A CO	+PAR1305
373	PAR1305 .0000 .0000 .0250 .0000 .0000 .0000	3
71C	2281305 - 8333 - 1.0000 - 0.	
375	3AR 1306 13 1.7275-1.3599-4 6.87375 1.43	+PAR1306
920	. 0000 .0050 .0000 .0000 .0050 .0000 .	130
770	P221306 ,8333 1.0000 0.	
g)	3AR 1307 13 .5242-1 .0175-4 1.20	+PAR1307
979	AR1307 .0000 .0000	.0000+P2R1307
080	221307 8333 1.0000 0.	
31	3AR 1303 13 1.0872-1.0352	AR 130
382	A21308 .0	+0000+P2R1308
83	221308 .8333 1.	
384	3.3	+PAR1309
385	A21309 .0000 .0100 .0100 .0000 .0000	28130
386	22R13098333 1.0000 0.	
787	A	2131
388	*AR1310 .0000 .0000 .0100 .0000 .0000 .0000	P2R13
986	>221310 .8333 1.0000 0.	
060	343 1311 13 .5650-1 .018	131
160	JAR1311 .0000 .0000 .0100 .0000 .0000 .0000	P2R13
392	>221311 .8333 1.0000 0.	
393	314 1312 13 .5630-1 .0138-4 .371	PAR131
560	2A21312 .0000 .0000 .0100 .0000 .0000 .0000	2813
960	22,1312 8333 1.0000 0.	
960	3.4	+PAR1313
97	. AR1313 .0000 .0000 .0100 .0000 .0000	28131
93	2281313 .8333 1.0000 0.	
99	∨ ∨	+PAR1314

3823-1 .0127-4 .11689 .0510-4 .0100 .0000 .0000 .0000 .4950-1 .0165-4 .28300 .0660-4 .0100 .0000 .0000 .0000 2	8333 1.0000 0.	28.1314	0.0 +PAR1315	0.0 .0000" .0000+P2R1316	.05" 2 +PAR2001	.21 .0 +0301	.11375 0.0 +0113	.140 +0216	•21 •0 +1145	•14 °0 +1149	.21	.043 0.0 +1011	.035 0.0 +T104	.04375 0.0 +1105	.0525 0.0 +1106	.06125 0.0 +1107	.070 0.0 +1108	75 0-0 +110	0.0 +111		111+ 0.0 6151	•035 0•0 +T204	.04375 0.0 +T205	.0525 0.0 +T206	106125 0 0 11301
3823-1 0127 3823-1 0127 0 0 100 0 0 1656 0 0 1656 0 0 165 0 0 163 1365 163 168 266 252 163 168 163 168 163 169 156 169 156 169 159 165 160 17055 163 160 156 160 156 163 163 165 160 165 165 165 165 165 165 165 165	8333 1.0000 0. 8333 1.0000 0. 113 2.05 163 113 2.252 163 113 2.084 2.252 163 104 0.042 154 107 0.050 156 108 0.050 156 109 0.055 159 109 0.055 163 109 0.055 163 109 0.055 163 109 0.055 163 109 0.055 163 109 0.055 155 109 0.055 155 109 0.055 155 109 0.055 155	AR 1315 AR 1315 AR 1315 AR 1315 AR 1315 AR 1315 AR 1316 AR 1316 AR 1316 AR 1316 AR 1316 AR 2000 AR 1318 AR 2001 AR 2001 AR 2000 AR	4 .11689 .051	4 .28300 .0660- 00 .0000 .00	-3.2333333.66567 0. 2.	3336-21	94-31	9514-310	3336-21	9514-31	6-21	41	-51	15-6502	0837-41	9-41	14-	6-41	0-41	,	101	-51	14-	0837-41	.33089-410
	8 8 3 3 3 1 3 1 3 1 3 1 3 1 3 1 3 1 3 1	2221314 .8333 1.2 2221315 .0000 2221315 .0000 2221315 .0000 2221316 .8333 1.2 2221316 .8333 1.3 222201 0.83333 0.8 222201 0.83333 0.8 222201 0.93333 0.8 222201 0.93333 0.8 222201 0.0 2222201 0.0 22222201 0.0 2222201 0.0 2222201 0.0 22222201 0.0 22222201 0.0 22222	.3823-1 .0127 .0100 .0	.4950-1 .0165 .0100 .0	.05 .1656	52 16	365 16	89	52 _ 16	68 16	52 16	50 11	42 154	0525 15	630 15	5 15	840 15	945 15	055 16	: "	5	45 24	525 15	630 15	35 15

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KDATA	· ·	.70326-410		.96469-410		.12840-310		.16670-310		.39514-310		.16955-210				64 88	35 36			63 87					
0 8 9 1	ŗ	159		260		261		292		266		163		112.2		05	12	103	112	39	ĸ	30	130	75.	
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SO	•	109	.04725	210	.0525	211	.05775	212	.063	216	.084	113	1365	112.1	2	5	15	84		3	2	2	2	•009	
	·	1209	•	1210	0	1211	0	1212	0	1215	0	1513	-0	131	2	3	15	83	33	130	10	- 11	12	009	
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	CARD	2151-	2152-	2153-	2154-	2155-	2156-	2157-	2153-		2150-	2161-	2162-	2163-	2164-	2165	2166-	2167-	2168-	2169-	2170-	-2171-	2172-	2173-	
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NO ERRORS FOUND - EXECUTE NASTRAN PROGRAM	
*** SYSTEM-INFORMATION-MESSAGE 3113, EMGPRO PROCESSING SINGLE PRECISION ELEMENTS OF TYPE34 STARTING WITH ID501	- 201
*** SYSTEM INFORMATION MESSAGE 3107. EMGOLD IS PROCESSING ELEMENTS OF TYPE = 34, BEGINNING WITH ELEMENT ID = 501	101
*** SYSTEM INFORMATION MESSAGE 3113, EMGPRO PROCESSING SINGLE PRECISION ELEMENTS OF TYPE 10 STARTING WITH ID	1
*** SYSTET INFORMATION MESSAGE 3113, ENGPROPROCESSING SINGLE PRECISION ELEMENTS OF TYPE 19 STARTING WITH ID 789 "	- 682
*** SYSTEM INFORMATION MESSAGE 3107. EMBOLD IS PROCESSING ELEMENTS OF TYPE = 19, BEGINNING WITH ELEMENT ID = 289	68
MESSAGE 3113, EYGPRO PROCESSING SINGLE PRECISION ELEMENTS OF TYPE 6 STARTING WITH ID	1
****SYSTEM"INFORMATION MESSAGE 3107.""EMGOLD'IS PROCESSING ELEMENTS OF TYPE = 6, BEGINNING WITH ELEMENT ID ** """""""""""""""""""""""""""""""""	1

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